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TECHNICAL NOTE 3473

EFFECTS OF SWEEP AND ANGLE OF ATTACK ON BOUNDARY-LAYER TRANSITION ON WINGS AT MACH NUMBER 4.04

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SUMMARY

Wind-tunnel tests were conducted at a Mach number of 4.04 to determine the effects of leading-edge sweep, angle of attack, and leading-edge thickness on boundary-layer transition on flat-plate wings. In addition, some results were obtained on wings having rounded leading edges. The transition point was determined for angles of attack up to 20° and for leading-edge sweep angles from 0° to 72° by the luminescent-lacquer technique of boundary-layer visualization.

The data showed that transition always occurred along a front parallel to the wing leading edge, and that increasing the leading-edge sweep angle or increasing the angle of attack between the undisturbed stream and model surface caused the transition line to move closer to the wing leading edge and, in general, decreased the transition Reynolds number.

An increase in the leading-edge thickness of a flat-plate wing with an unswept leading edge from 1/4 mil to 6 mils caused large increases in the local normal transition Reynolds numbers. However, on wings with 45° and 60° leading-edge sweep, increasing the leading-edge thickness had no apparent effect on the local normal transition Reynolds number.

A comparison of the NACA 65A004 section with the flat-plate section indicated that for small angles of leading-edge sweep the favorable pressure gradient due to the curved profile of the NACA 65A004 section produced longer lengths of laminar flow, and that for larger sweep angles the destabilizing effect of the curved streamline outside the boundary layer caused transition sooner than on the flat plate.

INTRODUCTION

The design of missiles and aircraft to fly at high supersonic Mach numbers requires a knowledge of boundary-layer conditions in order to predict heat-transfer effects and friction-drag coefficients. Much work has been done to determine the transition point on bodies of revolution and unswept surfaces at zero angle with the stream. For supersonic configurations, however, swept leading-edge surfaces operating at angles of attack are common, and the question arises as to the effects of sweep and angle of attack on boundary-layer transition. Very little theoretical or experimental work has been done on these problems at supersonic speeds. One of the earliest considerations of the sweep effect was the work of Jones (ref. 1) in which he considered the effects of sweep on two-dimensional compressible flow. He concluded that boundary-layer transition on swept wings would be influenced only by the component of velocity in the direction normal to the leading edge, and could therefore be treated as two-dimensional transition of the viscous or Tollmien-Schlichting type.

The purposes of the present investigation were (1) to determine the effects of sweep on boundary-layer transition on flat-plate wings at supersonic Mach numbers, (2) to investigate the effects of angle of attack on boundary-layer transition on such wings, (3) to determine to a limited extent the effects of wing profile on boundary-layer transition on swept wings at supersonic Mach numbers, and (4) to investigate the effects of leading-edge thickness on boundary-layer transition on flat-plate wings at supersonic Mach numbers. Data were obtained at various angles of attack in the Langley 9- by 9-inch Mach number 4 blowdown jet on flat and curved-surface wings having leading-edge sweeps from 0° to 72°.

A major consideration in any study of boundary layers in wind tunnels is the large possible effect of stream turbulence on the measured characteristics (ref. 2). However, in the light of the almost complete absence of data on this subject, the results of the present investigation, made at a constant free-stream turbulence level, are presented on the supposition that for other turbulence levels the effects of sweep and angle of attack may be qualitatively the same as those shown here although the values of the transition Reynolds number may be quite different.

SYMBOLS

l_{T_N} average normal distance from wing leading edge to transition front along central part of wing semispan

M_N calculated component of local Mach number just outside boundary layer, normal to wing leading edge

v_N calculated local velocity just outside boundary layer, normal to wing leading edge

T_2 calculated local static temperature just outside boundary layer

μ calculated local viscosity based on T_2

ρ	calculated local density based on T_2 and p_2
p_2	calculated surface static pressure
p_o	settling-chamber stagnation pressure
p_o'	equivalent stagnation pressure, $p_2\left(1 + \frac{\gamma' - 1}{2} M_N{}^2\right)^{\frac{\gamma'}{\gamma' - 1}}$
R_{T_N}	local transition Reynolds number normal to wing leading edge, $\frac{\rho v_N l_{T_N}}{\mu}$
R_{T_c}	calculated local transition Reynolds number normal to wing leading edge, based on an equivalent stagnation pressure other than that at which tests were made, $R_{T_N} + \Delta p_o' \frac{dR_{T_N}}{dp_o'}$
T_o	stagnation temperature, absolute
T	model temperature, absolute
T_s	stream static temperature, absolute
Λ	leading-edge sweep
θ_S	angle of attack between surface and stream (positive θ_S indicates compression and negative θ_S indicates expansion of the flow)
γ	trailing-edge sweep
γ'	ratio of specific heats (1.400 for air)
t	wing thickness
c	wing chord
h	wing trailing-edge thickness
b	wing span

Subscripts:

t	at wing tip
r	at wing root
max	maximum value

APPARATUS

The tests were conducted in the Langley 9- by 9-inch Mach number 4 blowdown jet. The settling-chamber pressure, which was held constant by a pressure-regulating valve, and the corresponding air temperature were continuously recorded during each run.

MODELS

The models used in these tests were wings of various plan forms, leading-edge sweeps, and sections (fig. 1). Plan forms were rectangular, arrow, trapezoidal, and delta. Leading-edge sweep angles varied from 0° to 72°. The airfoil sections used in the wings were of two general types - those having flat surfaces and sharp leading edges with double-wedge or hexagonal sections, and those having NACA 0003-63 and 65A004 subsonic-type sections. Most of the wings were semispan models mounted from a boundary-layer bypass plate as shown in figure 2. There were also six sting-supported wings, four with NACA 65A004 sections (wings 12 to 15), one having a double-wedge section and 60° leading-edge sweep (wing 8), and one having a half-circular-arc section and a square plan form with a 4-inch chord (wing 9).

All of the models were made of steel and had a root-mean-square surface roughness which ranged from about 5 to 25 microinches as measured by a profilometer (Physicists Research Co., Model No. 11). The leading-edge thicknesses of three of the sharp-leading-edge models (wings 1, 6, and 9) were varied from 1/4 mil to 10 mils by grinding the wing leading edges in planes perpendicular to the wing-chord plane and parallel to the leading edge. The leading-edge thicknesses of the other sharp-leading-edge models varied from 2 to 3 mils.

TESTS

The semispan models were mounted through a tunnel-wall boundary-layer bypass plate shaped to preserve the basic tunnel flow without introducing detrimental disturbances and located far enough from the tunnel wall so as to eliminate tunnel-wall boundary-layer effects. Because of adverse effects from choking behind the bypass plate at high angles of attack, the angle-of-attack range was limited to $\pm 14^{\circ}$. The sting-mounted models were tested in the center of the tunnel with their trailing edges far enough forward of the sting support so that interference from the support was considered negligible. Of all the sting-supported models, only wing 9 was tested at angles of attack, and it was tested from 0° to 20°. The transition points were made visible by means of the luminescent-lacquer technique (ref. 3), and then photographs were taken or direct measurements made.

For all the present tests, the settling-chamber stagnation temperature during a run varied from 80° to 60° F, and the settling-chamber stagnation pressure was 196 lb/sq in. abs, which corresponds to an undisturbed stream Reynolds number per foot of about 19.4×10^{6}. The tests were run at humidities below 5×10^{-6} pounds of water vapor per pound of dry air, which is believed to be low enough to eliminate water-condensation effects. The test-section static temperature and static pressure did not reach the point where liquefaction of air would take place.

PRECISION OF DATA

The probable accuracy of the measured quantities is summarized in the following table:

l_{T_N}, in.	± 0.1
T_o, deg	± 1
θ_S, deg	± 0.1
p_o, lb/sq in. abs	± 1

RESULTS AND DISCUSSION

Factors Which Can Affect Boundary-Layer Transition

In any investigation of boundary-layer phenomena, all the factors which can affect the characteristics of the boundary layer must, of course, be considered. For the present investigation these factors are the effects on the wing boundary layer of disturbances from the juncture of the wing leading edge and the boundary-layer bypass plate, effects on the boundary layer of the method used to determine boundary-layer transition, model surface condition, stream turbulence level, heat transfer, leading-edge thickness and related effects, leading-edge sweep, wing profile, angle of attack, Mach number, and Reynolds number.

It was decided to investigate systematically the effects on boundary-layer transition of leading-edge thickness, leading-edge sweep, angle of attack, and, to a limited extent, wing profile. The investigation of airfoil profile effects was confined to flat plates and NACA 0003-63 and NACA 65A004 sections (models 10 to 15). The other variables either could not be changed (stream Mach number), or were purposely held constant (stream turbulence level). The control or the effect of each variable will be discussed in the following sections after several general observations are made regarding the test results and the method of obtaining the data.

Figures 3 and 4 present representative series of photographs of the flow patterns on the wings made visible by luminescent lacquer. The darker regions along the wing leading edges are regions of laminar boundary-layer flow where the lacquer has not dried and, consequently, does not luminesce brightly. It will be noted that transition on these wings always occurred on a front parallel to the wing leading edge and upstream of the wing ridge lines except at the wing tip. It was assumed that there were no effects of the flow expansion around the ridge line on transition upstream of the ridge line, and that the data are representative of data obtained on swept flat-plate surfaces. To confirm this point, however, tests were made on wings having the same leading-edge sweep but different lengths of flat-plate section before the ridge line (wings 5 and 7) and on a wing having no ridge line (opposite surface of wing 5). Comparisons of l_{T_N}, the distance from the leading edge to boundary-layer transition, made at the same value of θ_S for all cases, revealed no difference in this length within the accuracy of the data.

The Effects of the Boundary-Layer Plate

The effects of the boundary-layer bypass plate on the wing boundary-layer flow pattern, as made visible by the luminescent lacquer, were investigated by testing a sting-mounted and a side-wall-mounted wing, both of which had 60° leading-edge sweep and a leading-edge thickness of about 2 mils (wings 7 and 8). No difference could be found between the lengths of laminar flow on the two wings at the same values of θ_S within the accuracy of the data, and it was therefore concluded that the method of support did not affect the data. Disturbances from the leading edge of the bypass plate did cause changes in the transition pattern near the apex of the delta wings (see figs. 3 and 4), but the measurements were made outboard of these areas to avoid these disturbances and also to avoid the conical-flow area from wing apexes.

The Luminescent-Lacquer Technique

The effect of the presence of the luminescent lacquer on the transition measurements was the object of some concern, and great care was taken to apply a uniform coat of the lacquer for each test. Data were not taken when the flow pattern showed any evidence of ripples, runs, or unevenness in the lacquer film. Measurements of the roughness of the surface of the lacquer could not, of course, be made. In reference 4, Lange and Gieseler used both the luminescent-lacquer and the spark-schlieren method to make measurements of boundary-layer transition on a slender cone at Mach numbers between 1.9 and 4.2. They concluded that the two methods agreed fairly well, the luminescent-lacquer measurements yielding Reynolds numbers of transition that were lower by about 0.2×10^6 for Reynolds numbers of transition between 1.5×10^6 and 3×10^6.

In reference 5, Potter concluded that the general agreement between the transition stations determined on cone-cylinder bodies at Mach numbers from 2.17 to 3.24 by the luminescent-lacquer method and skin-friction measurements is satisfactory. Considering the evidence of references 4 and 5 and the experience with the luminescent-lacquer technique in the present investigation, no good reasons can be found why the use of the luminescent lacquer should invalidate the trends of boundary-layer transition-point movement with leading-edge sweep, angle of attack, and leading-edge thickness presented in this paper.

Model Surface Condition

Measurements of the model surface roughness gave root-mean-square values of 5 to 25 microinches. Such a variation at this low level of roughness would not be expected to influence boundary-layer transition. Tests of wings with the same sweep and leading-edge thickness and the smallest and largest roughnesses measured did not indicate any differences in the Reynolds number for transition. The models were, of course, cleaned with alcohol and resprayed with lacquer between runs. The lacquer coating might well mask out any effects of the measured variations in surface roughness.

Stream Turbulence Level

The Reynolds numbers for transition on an ogival nose and on a flat plate with a 0.0025-inch-thick leading edge tested at a stagnation pressure of 196 lb/sq in. abs in this facility are about 2×10^6, which may indicate a rather high stream turbulence level. However, the stream turbulence level was maintained constant for all tests by holding the stagnation pressure and temperature constant (within $\pm 10^{\circ}$).

Heat-Transfer Effects

Heat transfer to or from the models could not be controlled in this investigation since no provision was made for the control of the air or model temperatures. To determine the approximate time required for the models to reach equilibrium temperature, a thermocouple was installed in one of the side-wall-mounted wings near the wing tip. Tests at zero angle of attack showed that equilibrium temperatures were reached after about 80 seconds' running time (see fig. 5). The transition patterns in the luminescent lacquer were formed after about 30 seconds and did not change when runs were extended to 2 minutes. At angles of attack the recovery temperatures probably differed on the upper and lower surfaces of the wing, causing heat transfer through the wing.

However, it will be shown that the boundary-layer-transition data obtained under the aforementioned conditions of heat transfer correlate on the basis of parameters which do not take heat transfer into account and that, therefore, heat-transfer effects are probably of secondary importance in this investigation. This conclusion is in agreement with the analysis of reference 2, which showed that when the transition Reynolds number for zero heat transfer was low, as was also the case in the present tests, the effects of heat transfer on boundary-layer transition were small.

Effects of Leading-Edge Thickness

Tests were made to determine the effects of leading-edge thickness on a rectangular sting-supported flat-surface wing of square plan form (wing 9) and on two side-wall-mounted delta wings having wedge airfoil sections and 45° and 60° leading-edge sweep (wings 1 and 6). The leading-edge thickness of the wing with rectangular plan form was varied from 1/4 mil to 6 mils. Increasing the leading-edge thickness produced longer laminar runs (fig. 6) and higher local transition Reynolds numbers (fig. 7). An increase in transition Reynolds number with leading-edge thickness on flat plates and open-nose cylinders was also found in references 6 and 7. The data of reference 7 showed an approximately constant rate of increase of transition Reynolds number with leading-edge thickness up to 12 mils, whereas the present data exhibited a negligible rate of increase of transition Reynolds number as the leading-edge thickness was increased from 4 to 6 mils.

Increasing the leading-edge thickness of the swept wings from 1 mil to 10 mils produced no changes in the location of the transition point at angles of attack up to 10°, within the accuracy of the measurements (fig. 8). Although no effects of leading-edge thickness on boundary-layer transition on the wings with swept leading edges could be determined, the leading-edge thicknesses were kept between 2 and 3 mils.

Effects of Leading-Edge Sweep and Angle of Attack

Increasing the leading-edge sweep angle of the flat-plate wings (figs. 3 and 9) or increasing the surface angle of attack θ_S (figs. 4 and 9) caused boundary-layer transition to move closer to the wing leading edge. This movement, in general, corresponded to a decrease in transition Reynolds number (fig. 10; increasing θ_S decreases M_N).

The values of local Mach number and transition Reynolds number plotted in figure 10 were obtained from theoretical calculations of the local static pressures and the Mach number components normal to the leading edge and from the experimental distances from the leading edge to the transition front (fig. 9). To determine the accuracy of the local-pressure calculations, pressures were measured on the forward surface of a wing with double-wedge section and 50° leading-edge sweep (fig. 11). The experimental surface pressures agreed with the theoretical pressures to within 5 percent of the theoretical value and indicated no pressure gradient on the forward surface of the wing at angles of attack from -8° to 14°.

Variations of Transition Reynolds Number With Pressure

Simple sweep theory (ref. 1) predicts that the local flow conditions on swept-wing surfaces, including boundary-layer transition, are a function only of the normal Mach number component. In the present tests, the same surface Mach number normal to the leading edge was obtained at several leading-edge sweep angles by changing the angle of attack (fig. 10), and for such conditions large variations of the local transition Reynolds numbers occurred. However, when the same surface Mach numbers normal to the leading edge were obtained, differences in surface pressure and temperature also existed. This fact, together with the observed disagreement of the data with the prediction of the sweep theory, suggests that perhaps, as has been shown in references 7 to 10 for flat plates and open-nose cylinders tested in wind tunnels, there is a regular variation of the transition Reynolds number with pressure or some parameter which is a function of pressure.

Therefore, in figure 12 the variation of the transition Reynolds number with calculated surface static pressure as determined from the present tests is presented for three local Mach numbers normal to the leading edge. It can be seen that the rate of change of transition Reynolds number with surface static pressure is linear and varies with Mach number. However, when the same data were plotted, using an equivalent stagnation pressure p_o' determined from the theoretical surface pressure and surface Mach number normal to the leading edge, the data show (fig. 13) that there is a linear increase of the transition Reynolds number with equivalent stagnation pressure which remains about constant for the range of local Mach numbers of the present investigation.

Since the transition Reynolds number of the present tests increased linearly with equivalent stagnation pressure, this rate of increase was used to adjust the data to one equivalent stagnation pressure (200 lb/sq in. abs). Transition Reynolds numbers obtained in this manner are presented in figure 14, and it is seen that the calculated transition Reynolds numbers are the same at the same local normal Mach number for wings of varying sweep. Thus, for these tests the effects of leading-edge sweep and angle of attack on boundary-layer transition have been correlated by considering the equivalent stagnation pressure and the rate of change of the transition Reynolds number with equivalent stagnation pressure. It can also be seen from figure 14 that there is still a change in transition Reynolds number with surface Mach number normal to the leading edge which appears similar to the trend obtained in tests of hollow cylinders at Mach numbers from 2.2 to 5.0 with constant stagnation pressure at the U. S. Naval Ordnance Laboratory (ref. 11). Whether or not these results yield a trend of the change in transition Reynolds number with Mach number that can be applied to flat plates at other free-stream Mach numbers in other wind tunnels is a point that will require further investigation.

Figure 13 also presents the rates of change of transition Reynolds number with stagnation pressure obtained in other wind-tunnel investigations on flat plates and open-nose cylinders at Mach numbers from 1.97 to 4.54 (refs. 7 to 10). An examination of these rates of increase shows that the rates do not vary by more than a factor of 2.5. The actual values of transition Reynolds number at a given stagnation pressure vary greatly among the investigations. This may be a result of the variation in turbulence levels in the different facilities and the differences in the models tested.

Effects of Changes in Wing Section

Five wings with NACA 65A004 sections and one with an NACA 0003-63 section were tested. The wings with the 65A004 section, which were designed for use in another wind tunnel, were tested only at zero angle of attack because of stress limitations. The test results, in terms of the normal distance from the wing leading edge to the boundary-layer transition front, which was parallel to the leading edge, are presented in figure 15. The local transition Reynolds number could not be computed because of the lack of a method to predict local conditions on such wings at Mach number 4.

Other investigations of boundary-layer transition on wings with unswept leading edges have demonstrated increased stability of the boundary layer due to the negative pressure gradient obtained on a convex surface (refs. 12 and 13). This effect was also indicated by wings 11 and 15 of the present tests. Wing 11, which had only 12.5° sweep, had a root chord of 4 inches and a tip chord of 2.11 inches and had laminar boundary layer over all of its surface except for small regions near the root chord and the wing tip. At sweep angles greater than about 30°, shorter runs of laminar boundary layer were obtained on the round-leading-edge wings than on the flat-surface wings. This is probably due to the increase in strength of the destabilizing effect on the boundary layer caused by curvature of the flow just outside of the boundary layer, which has been demonstrated on swept wings at subsonic speeds by J. T. Stuart and W. E. Gray in England.

SUMMARY OF RESULTS

An analysis has been made of the effects of leading-edge sweep, surface angle of attack, and leading-edge thickness on the movement of boundary-layer transition on wing surfaces in a wind tunnel at Mach number 4.04 in terms of the components of local Mach number and local Reynolds number normal to the wing leading edge. The following results were obtained:

1. Transition always occurred along a front parallel to the wing leading edge, and increasing the leading-edge sweep angle or increasing the angle of attack between the undisturbed stream and the model surface caused the transition front to move closer to the wing leading edge and decreased the local normal transition Reynolds number.

2. These tests give a linear and equal rate of increase of transition Reynolds number with equivalent stagnation pressure (stagnation pressure calculated from the local static pressure and the local Mach number normal to the leading edge) at all local normal Mach numbers. This rate is in general agreement with the rate of increase of transition Reynolds number with actual stagnation pressure obtained in other wind-tunnel investigations on flat plates and open-nose cylinders at Mach numbers from 1.97 to 4.54.

3. A correlation has been obtained, for the present data, of the effects of leading-edge sweep and angle of attack on boundary-layer transition on plane surfaces when the equivalent stagnation pressure and the rate of change of the transition Reynolds number with equivalent stagnation pressure are considered.

4. Increasing the leading-edge thickness of a flat-plate wing with an unswept leading edge from 1/4 mil to 6 mils caused large increases in the local normal Reynolds number for boundary-layer transition.

5. Increasing the leading-edge thickness of wings with 45° and 60° leading-edge sweep from 1 to 10 mils produced no changes in the local normal Reynolds number for boundary-layer transition.

6. For small angles of leading-edge sweep, a 65A004 section had longer lengths of laminar flow than the flat-plate section, but for large angles of leading-edge sweep, shorter lengths of laminar flow were obtained.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., March 2, 1955.

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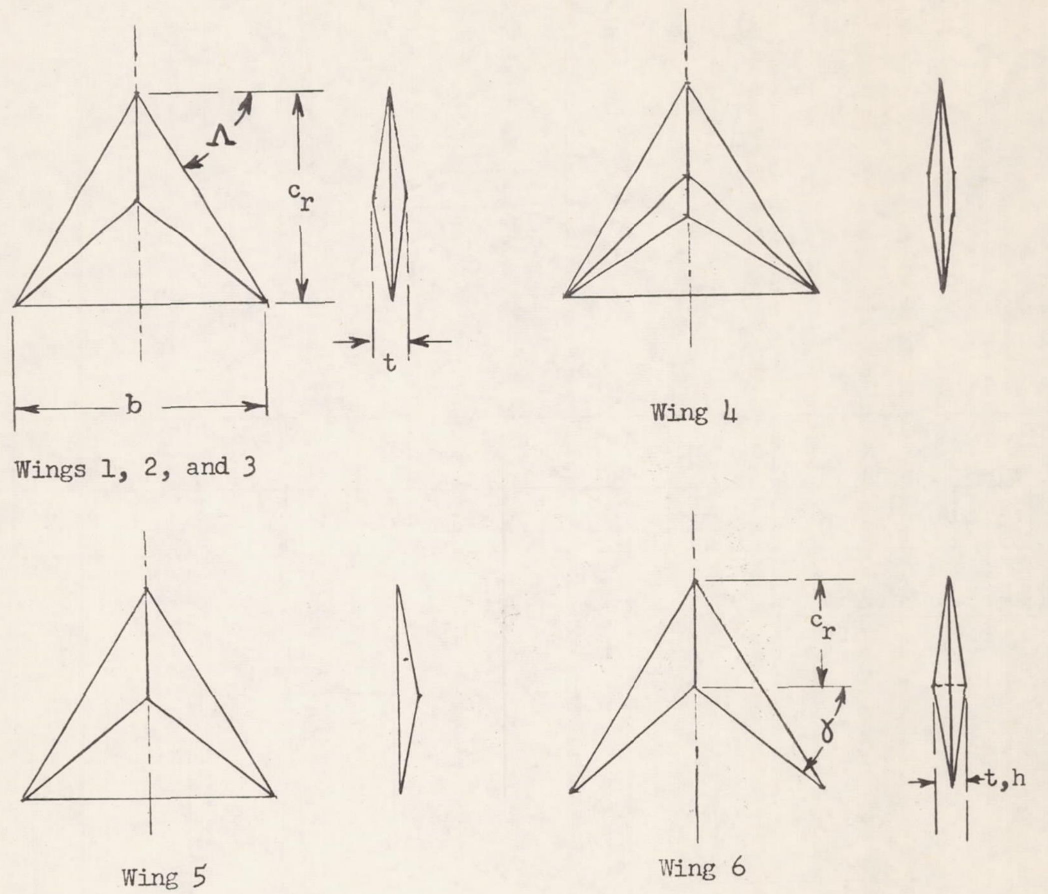
SHARP-LEADING-EDGE WINGS

Side-wall mounted.- Tested as half wings.

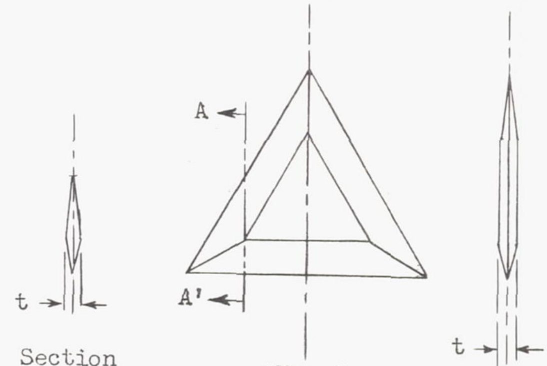


Wing	Λ (deg)	γ (deg)	Aspect Ratio	$(t/c)_{max}$	$(t/c)_{max}$ location	h/t	c_r (in.)	b (in.)
1	45	0	4.00	0.05	0.50c	0	4.99	9.98
2	55	0	2.80	.05	.50c	0	4.99	6.99
3	72	0	1.30	.08	.50c	0	5.98	3.89
4	50	0	3.36	.0395	.40—.60c	0	4.99	8.38
5	60	0	2.31	.05	.50c	0	9.00	10.39
6	60	33.6	3.87	.13	1.00c	1.0	3.52	6.80

Figure 1.- Wing dimensions.

SHARP-LEADING-EDGE WINGS

Side-wall mounted.- Tested as half-wings.

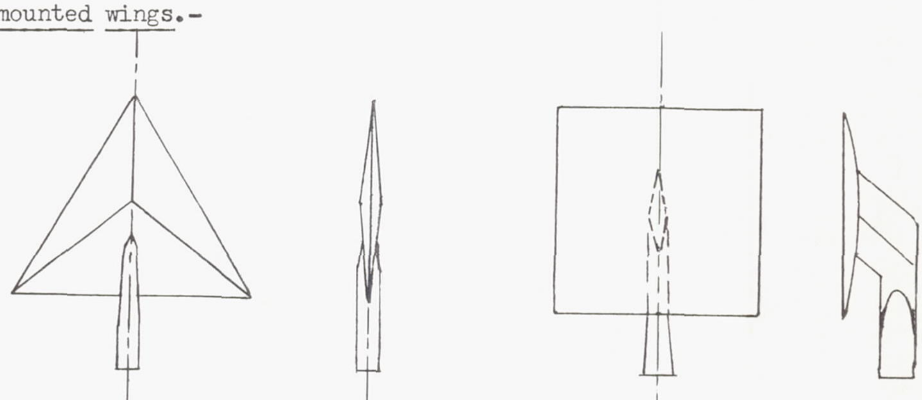


Section through A-A'

Wing 7

Wing	Λ (deg)	γ (deg)	Aspect Ratio	$(t/c)_{r_{max}}$	$(t/c)_{r_{max}}$ location	$(t/c)_{t_{max}}$	$(t/c)_{t_{max}}$ location	h/t	c_r (in.)	b (in.)
7	60	0	2.31	0.04	.30—.87c_r	0.0922	0.692 c_t	0	5.93	6.85

Sting-mounted wings.-



Wing 8

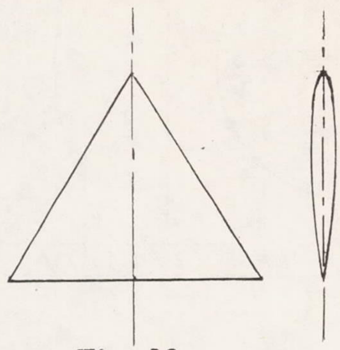
Wing 9

Wing	Λ (deg)	γ (deg)	Aspect Ratio	$(t/c)_{max}$	$(t/c)_{max}$ location	Section	h/t	c_r (in.)	c_t (inc.)	b (in.)
8	60	0	2.31	0.05	0.50c	Double wedge	0	4.36	0	5.03
9	0	0	1.00	0.05	0.50c	Half-circular-arc	0	4.00	4.00	4.00

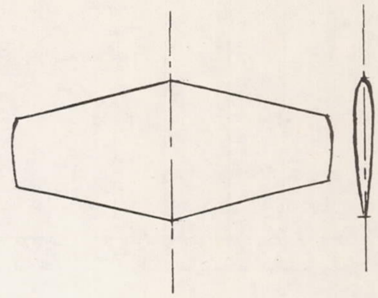
Figure 1.- Continued.

ROUNDED-LEADING-EDGE WINGS

Side-wall mounted.- Tested as half-wings.



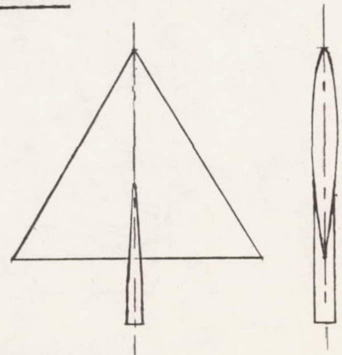
Wing 10



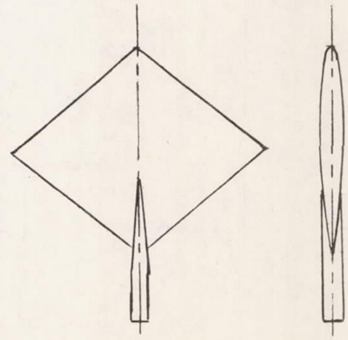
Wing 11

Wing	Λ (deg)	γ (deg)	Aspect Ratio	$(t/c)_{max}$	Section	c_r (in.)	c_t (in.)	b (in.)
10	63.4	0	2.0	0.03	NACA 0003-63	6.0	0	6.0
11	12.5	-12.5	3.0	0.04	NACA 65A004	4.22	2.11	9.48

Sting-mounted.-



Wings 12 and 13



Wings 14 and 15

Wing	Λ (deg)	γ (deg)	Aspect Ratio	$(t/c)_{max}$	Section	c_r (in.)	c_t (in.)	b (in.)
12	60	0	2.31	0.04	NACA 65A004	4.74	0	5.48
13	45	0	4.00	0.04	NACA 65A004	3.60	0	7.20
14	41	-41	2.31	0.04	NACA 65A004	4.74	0	5.48
15	27	-27	4.00	0.04	NACA 65A004	3.60	0	7.20

Figure 1.- Concluded.

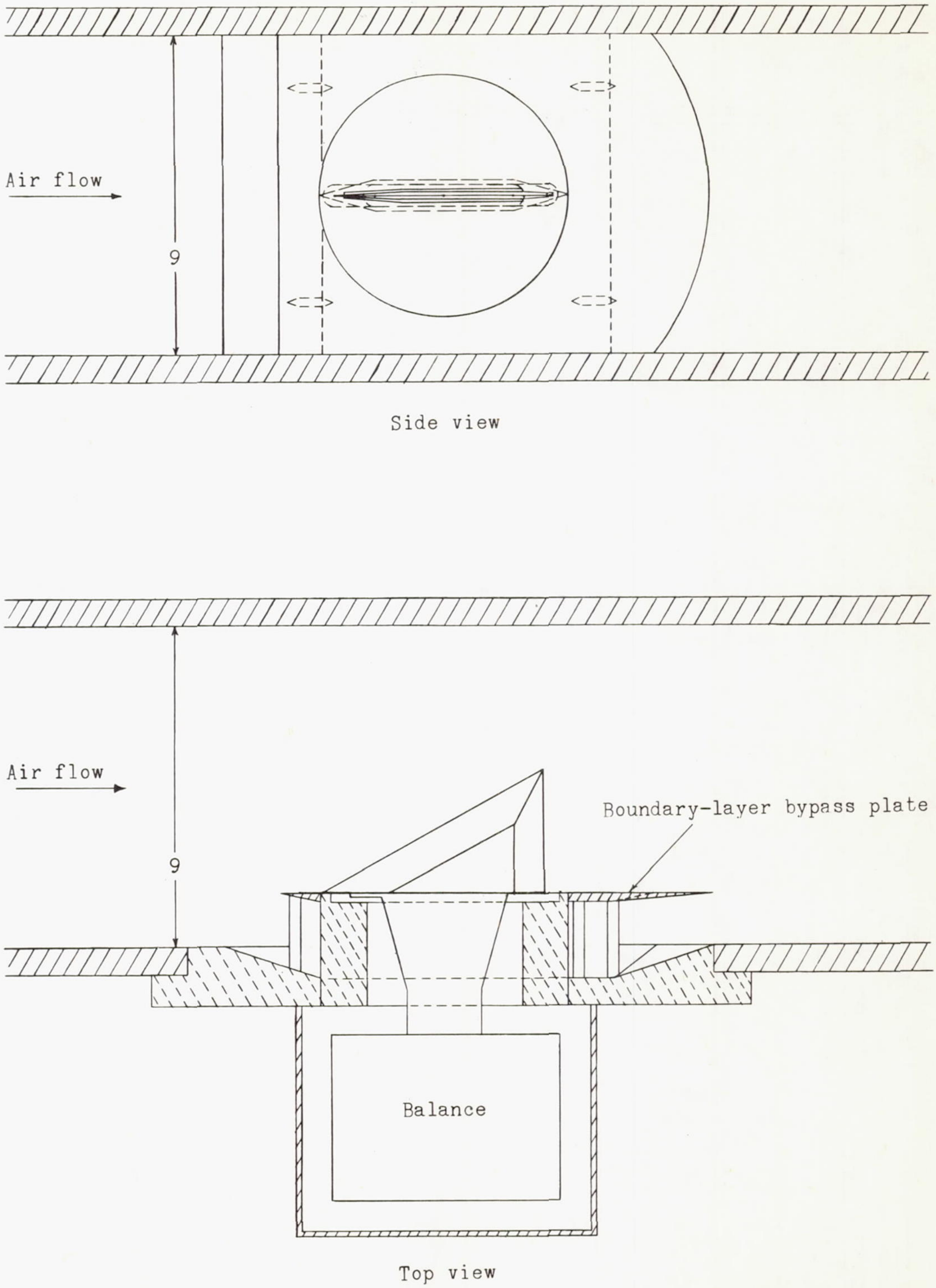
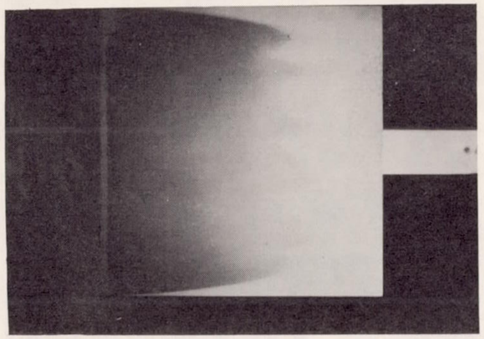
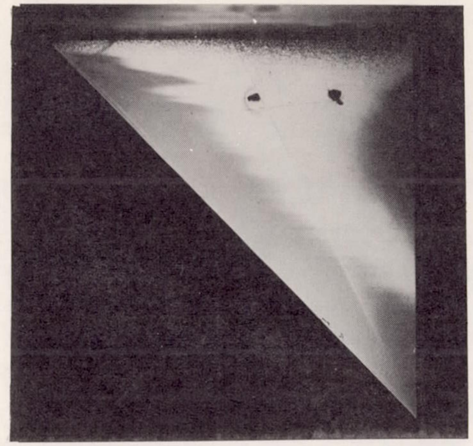


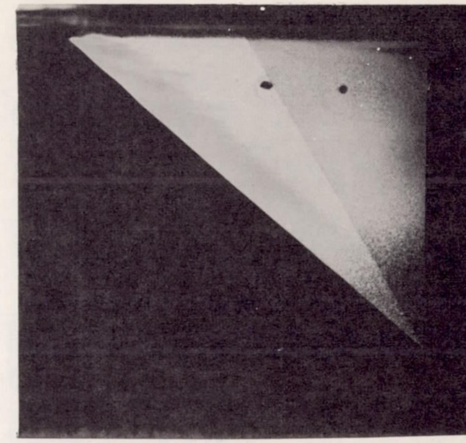
Figure 2.- Schematic diagram of test section of Langley 9- by 9-inch Mach number 4 blowdown jet and balance arrangement. Dimensions are in inches.



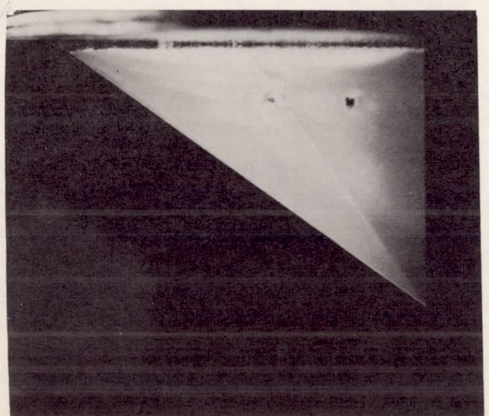
$\Lambda = 0^{\circ}$



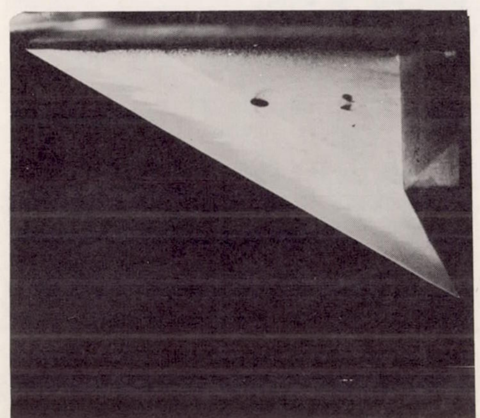
$\Lambda = 45^{\circ}$



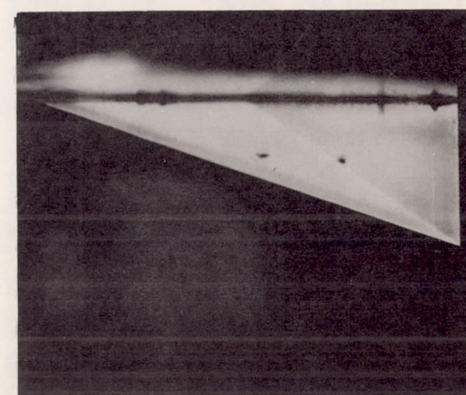
$\Lambda = 50^{\circ}$



$\Lambda = 55^{\circ}$



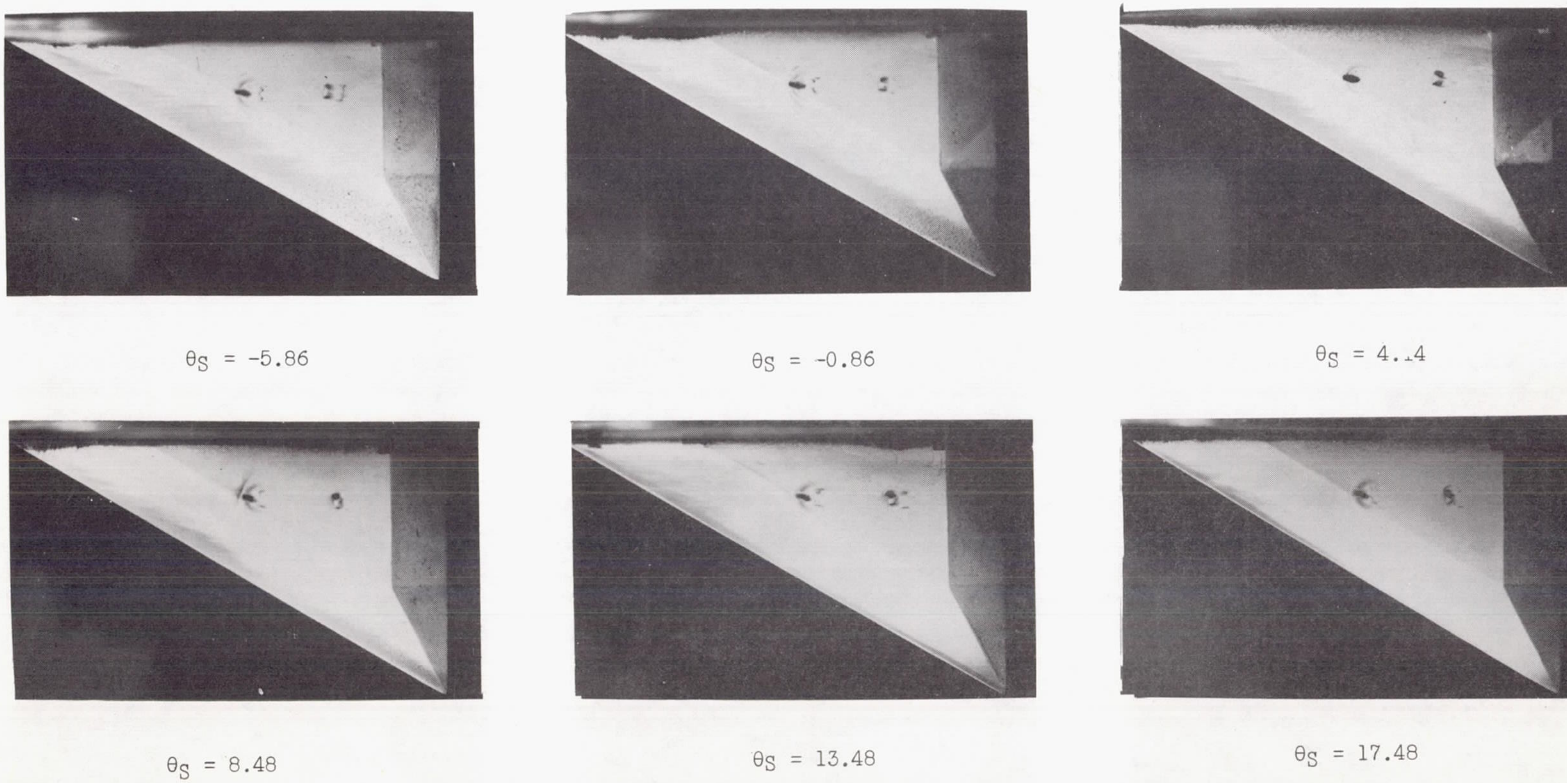
$\Lambda = 60^{\circ}$



$\Lambda = 72^{\circ}$

L-86443

Figure 3.- The effects of leading-edge sweep angle on boundary-layer transition. $M = 4.04$.



L-86444

Figure 4.- The effects of angle of attack on boundary-layer transition. M = 4.04; wing 7.

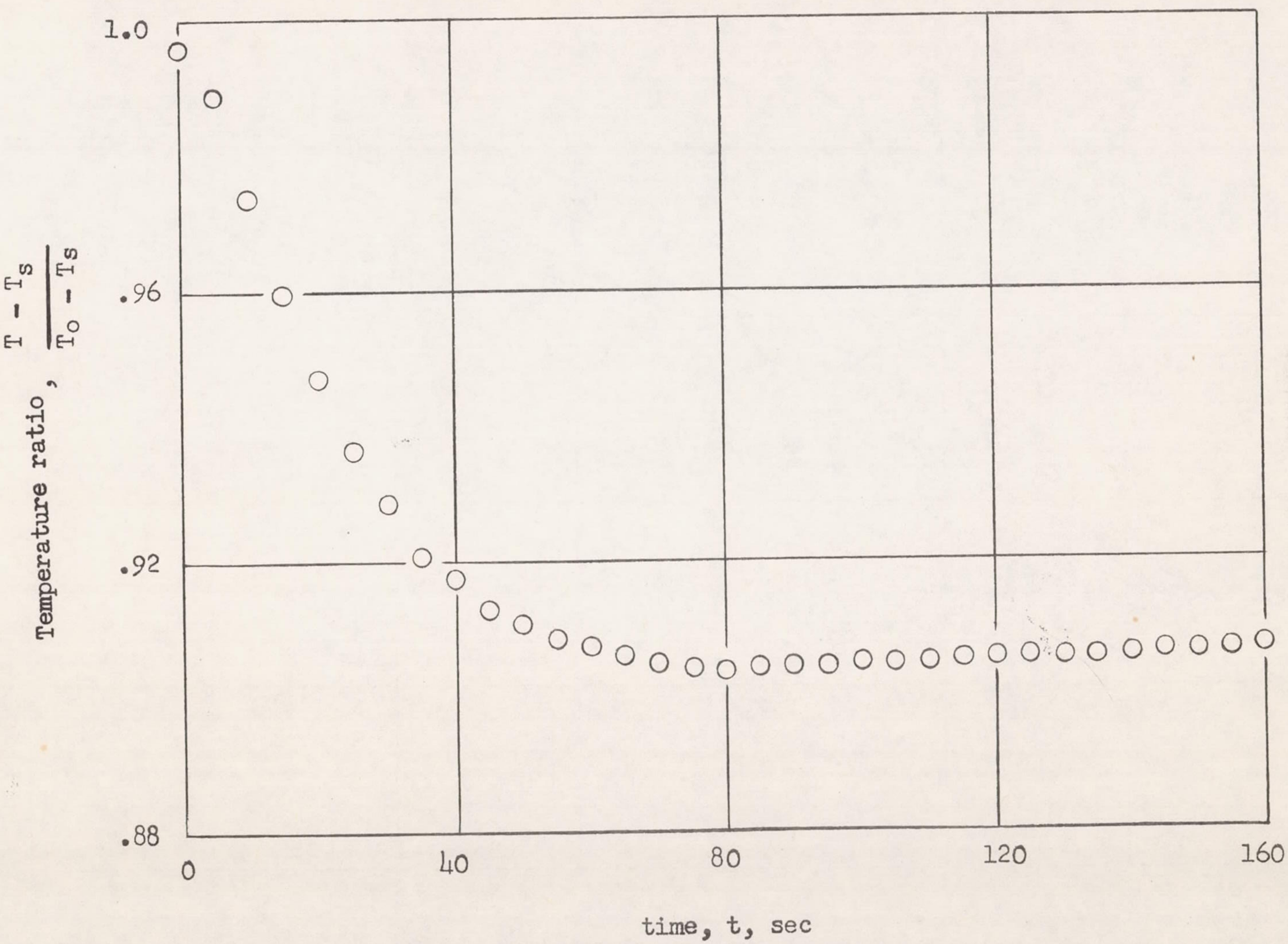


Figure 5.- Variation of wing temperature with time. M = 4.04; wing angle of attack, 0°; leading-edge sweep, 60°.

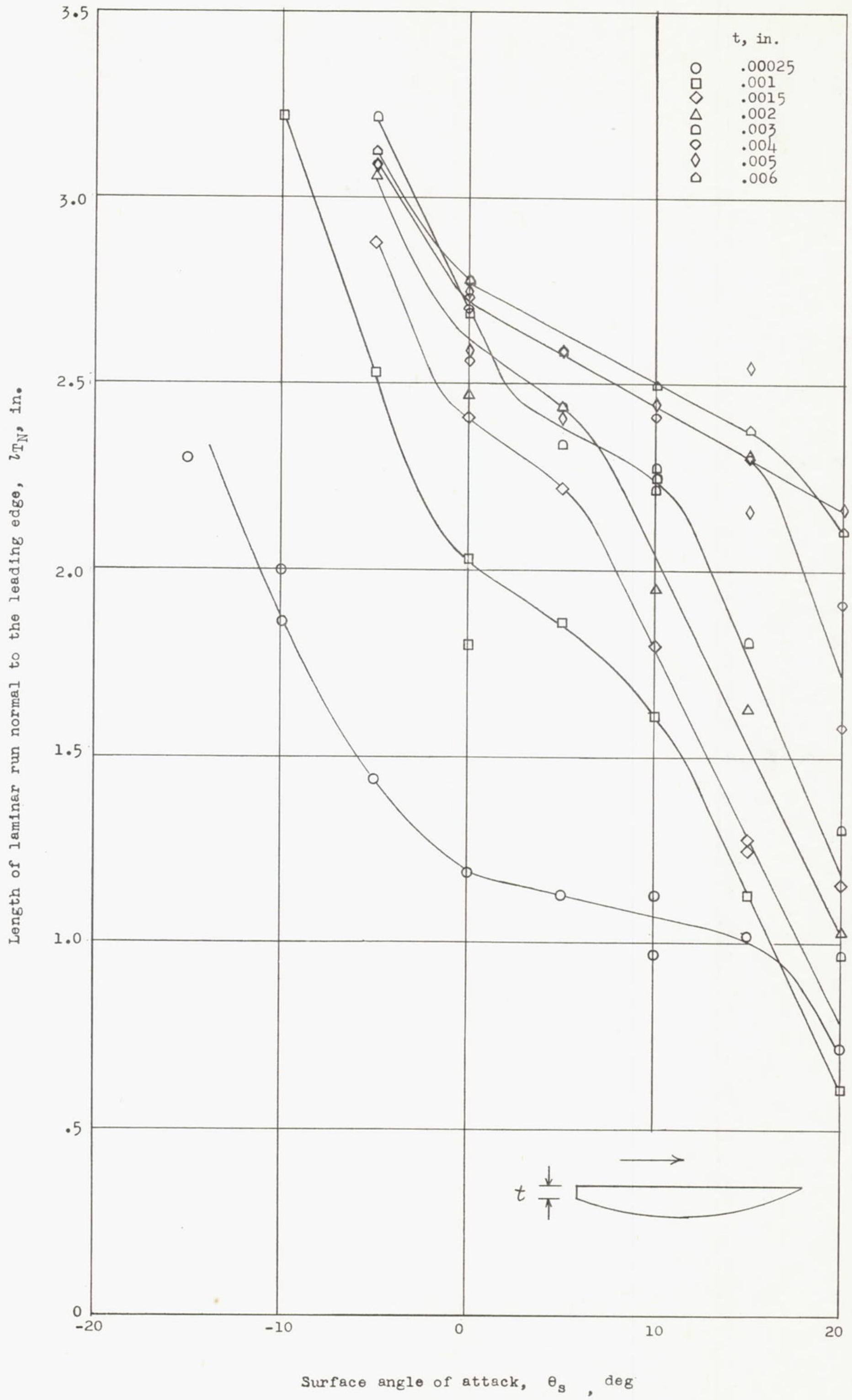


Figure 6.- Effects of leading-edge thickness on boundary-layer transition on a wing with rectangular plan form. M = 4.04.

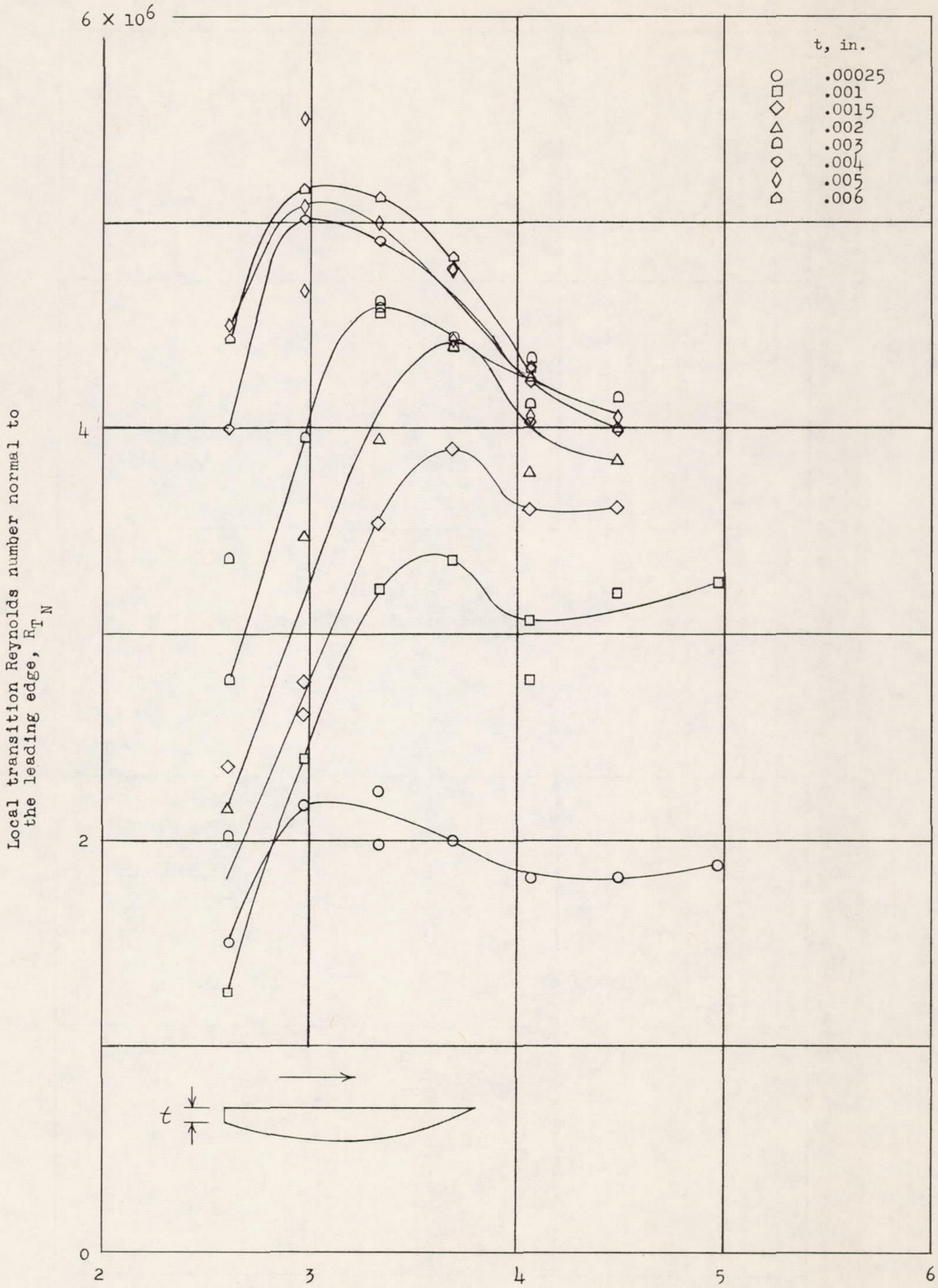
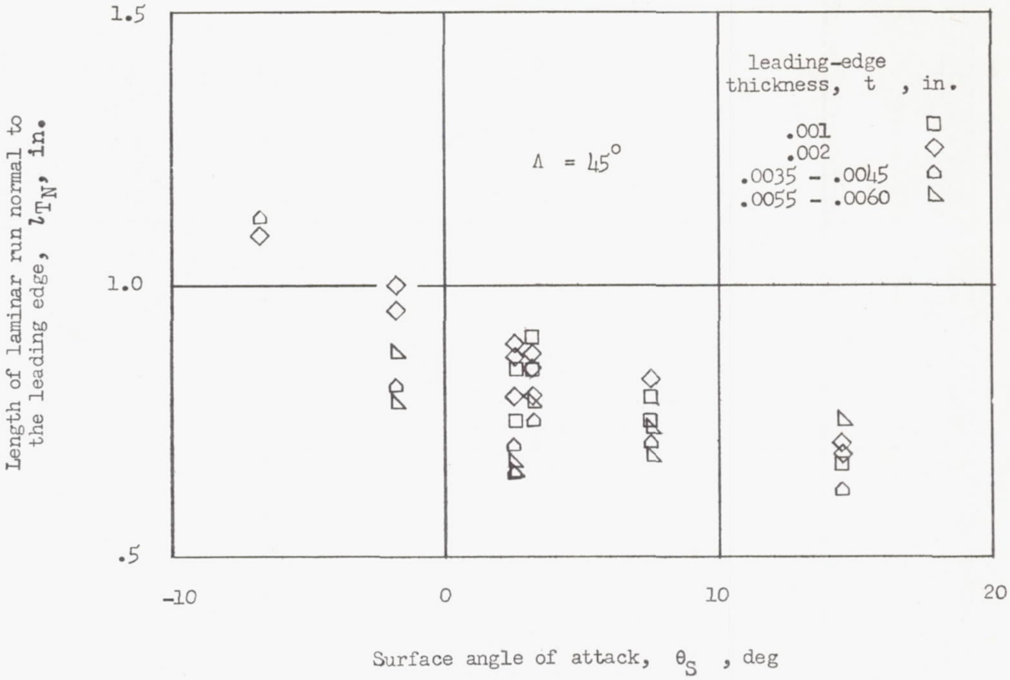


Figure 7.- Effects of leading-edge thickness on the variation of local transition Reynolds number with surface Mach number on a wing with rectangular plan form. M = 4.04.



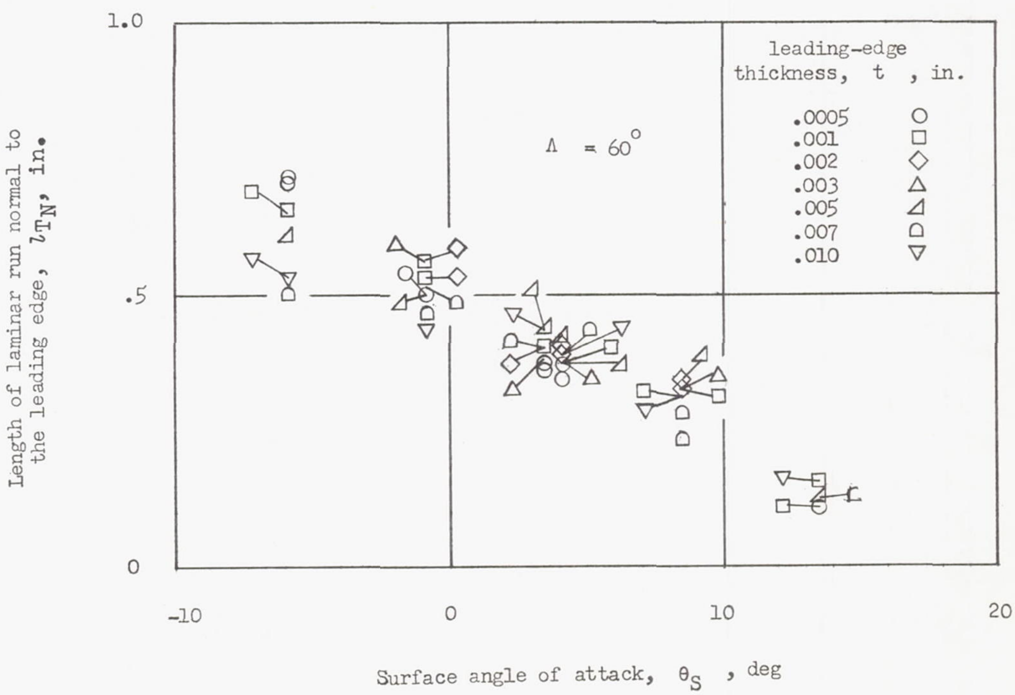


Figure 8.- Effects of leading-edge thickness on boundary-layer transition on flat-surface wings with swept leading edges. M = 4.04.

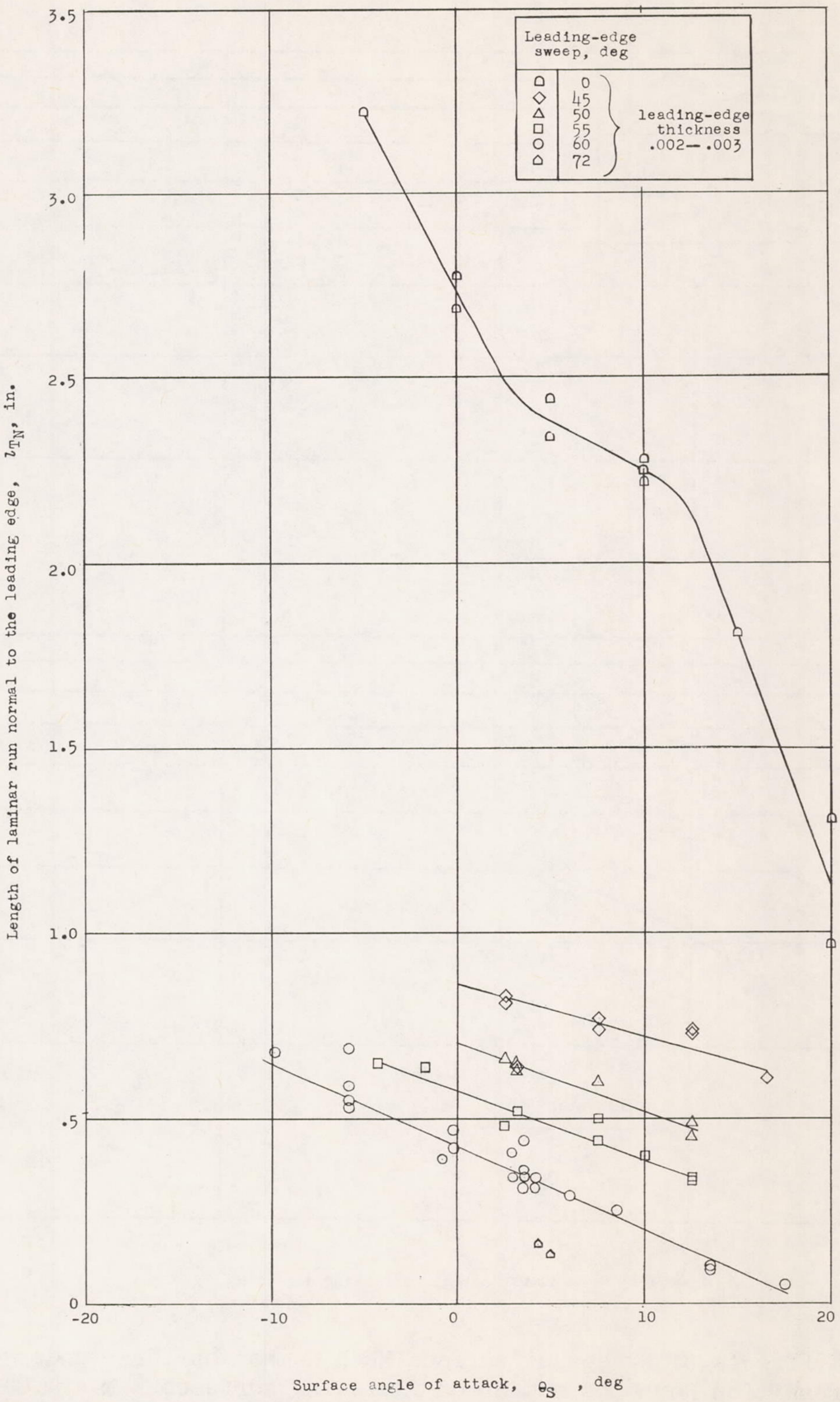


Figure 9.- Effects of sweep and angle of attack on the location of boundary-layer transition on flat-wing surfaces. M = 4.04.

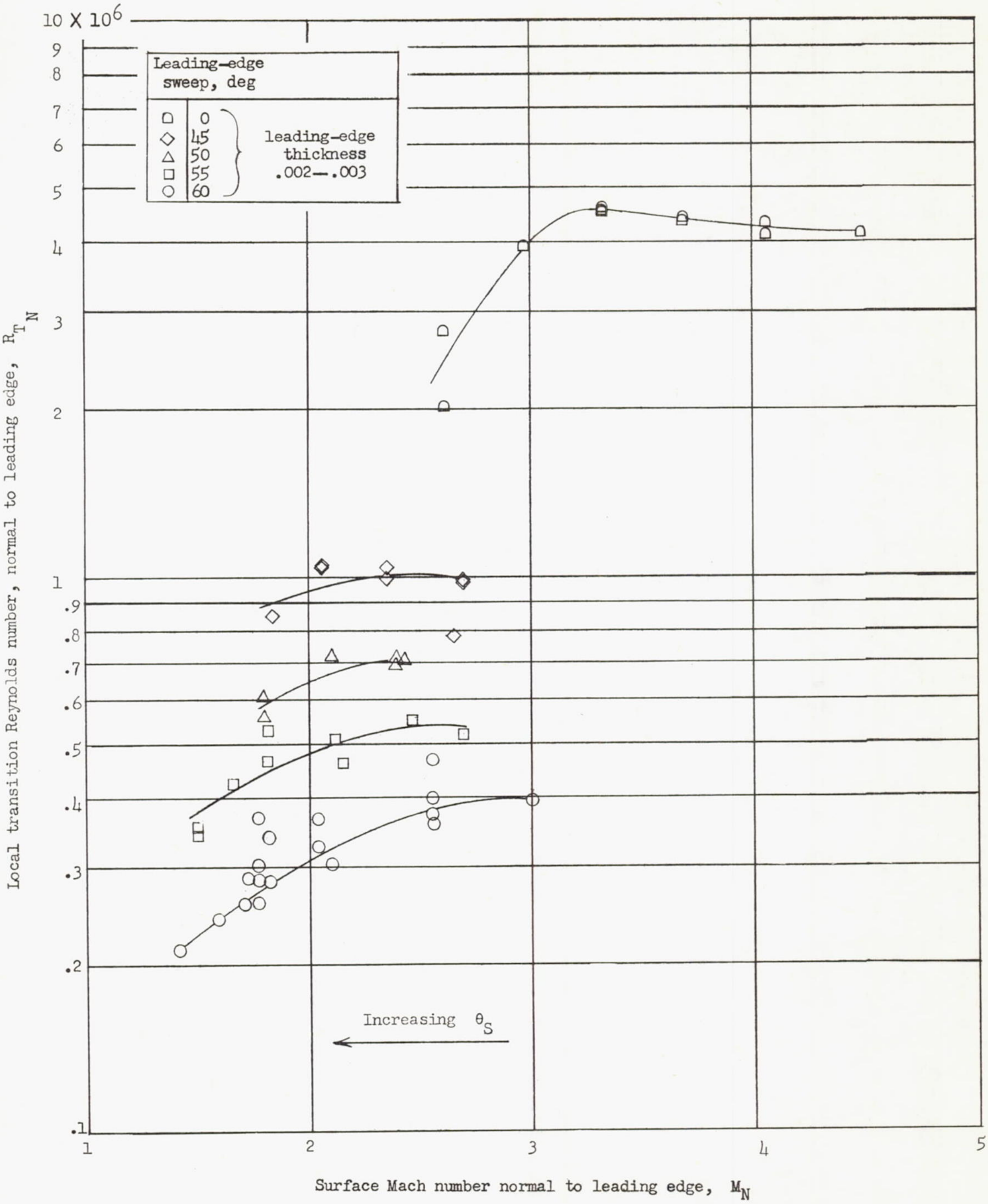


Figure 10.- Effects of sweep and surface Mach number on the boundary-layer transition Reynolds number on flat-wing surfaces. M = 4.04.

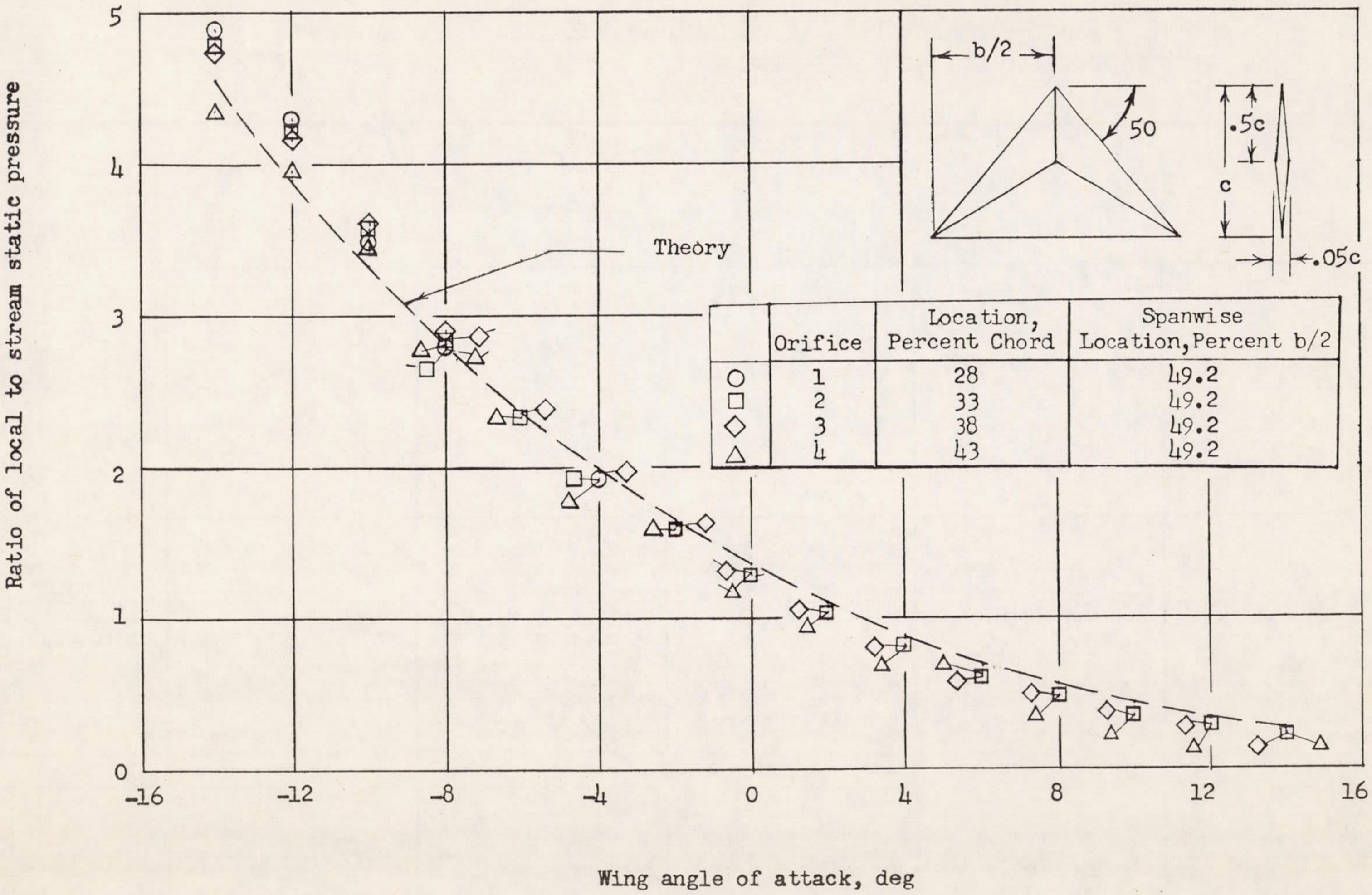


Figure 11.- Comparison of theoretical and experimental pressures on the upper surface of the forward wedge of a double-wedge-section delta wing with 50° leading-edge sweep. $M = 4.04$.

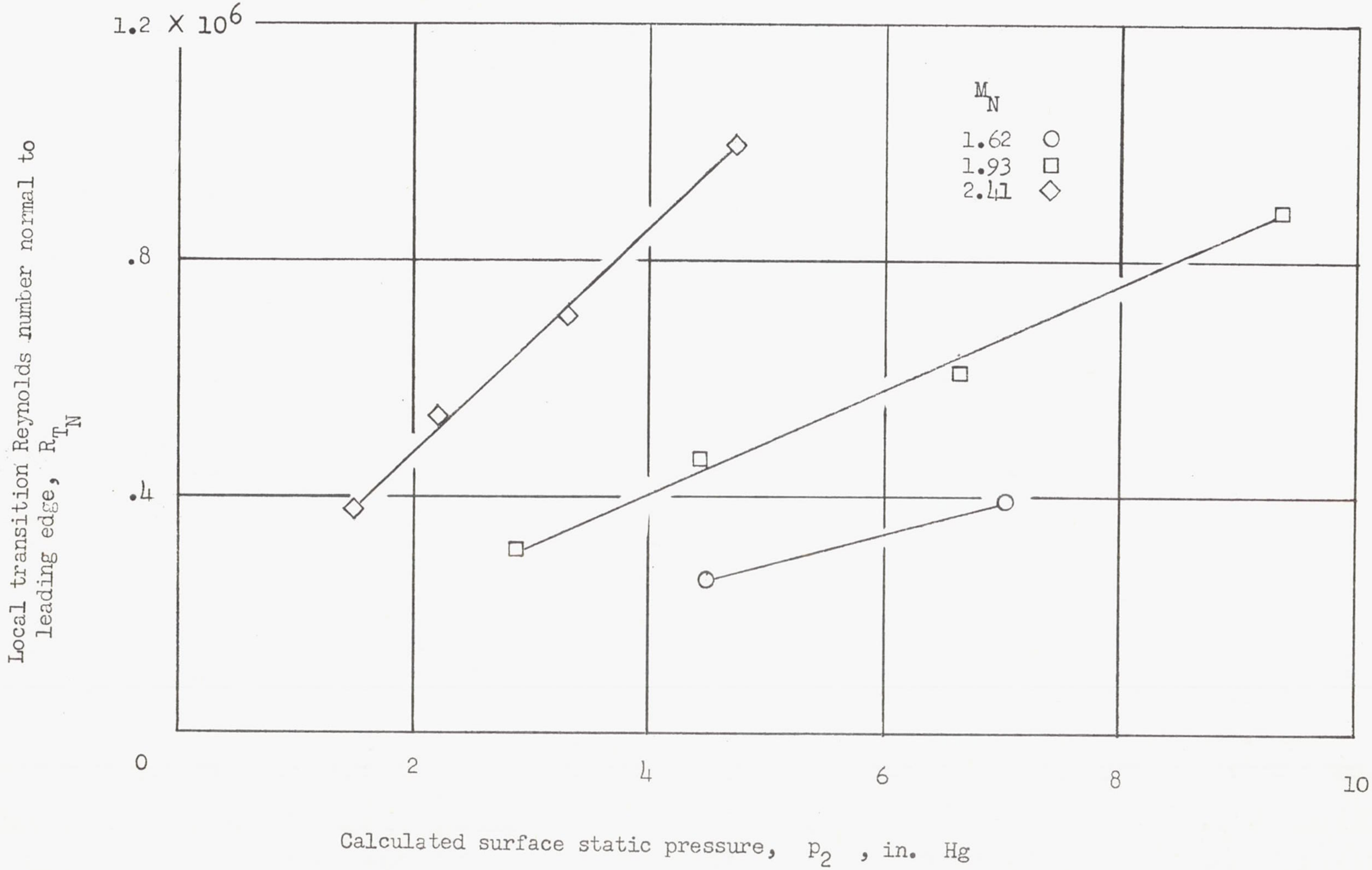


Figure 12.- Variation of the local normal transition Reynolds number with surface static pressure on swept-wing surfaces. M = 4.04.

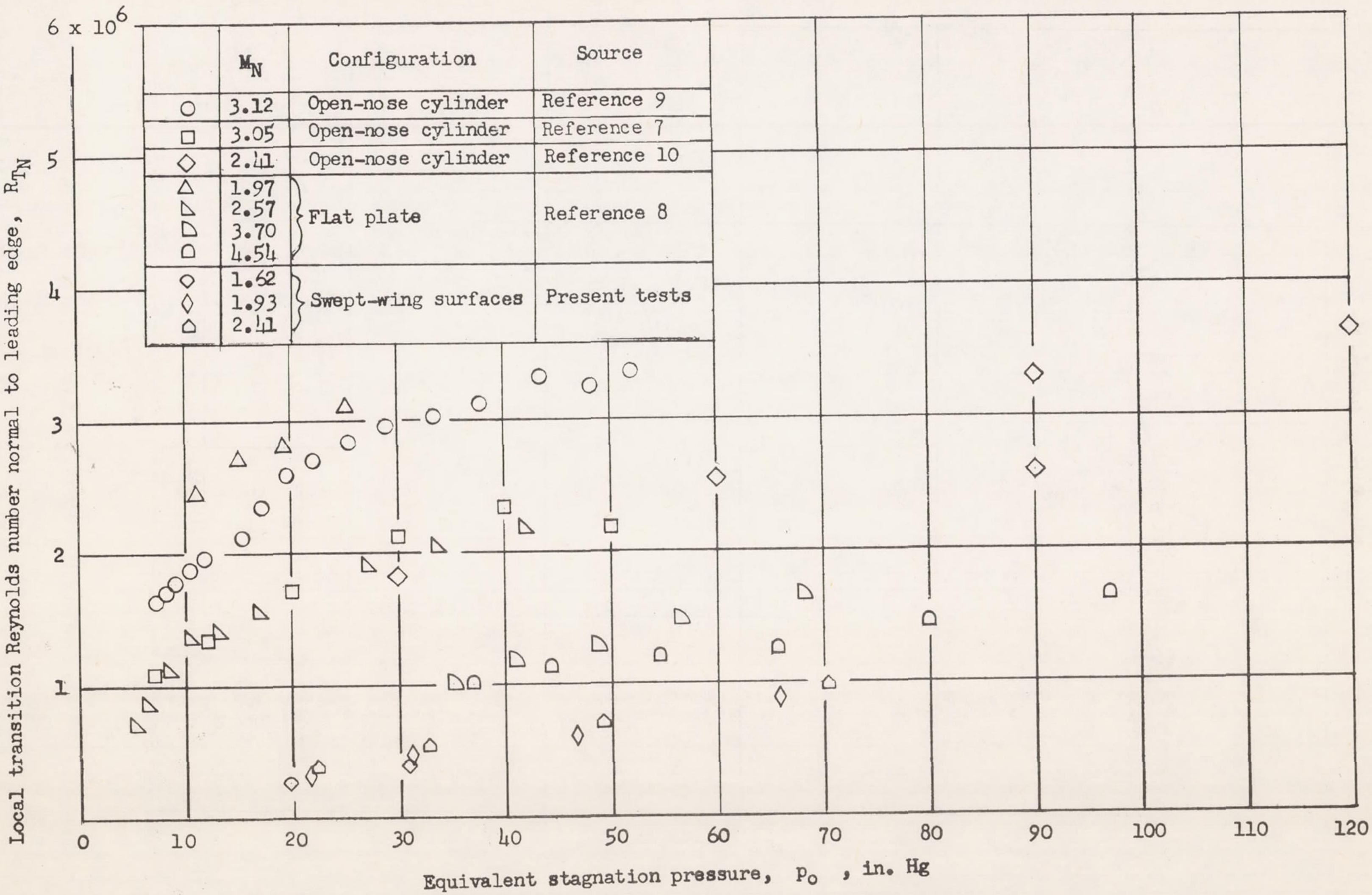


Figure 13.- Variation of transition Reynolds number with stagnation pressure.

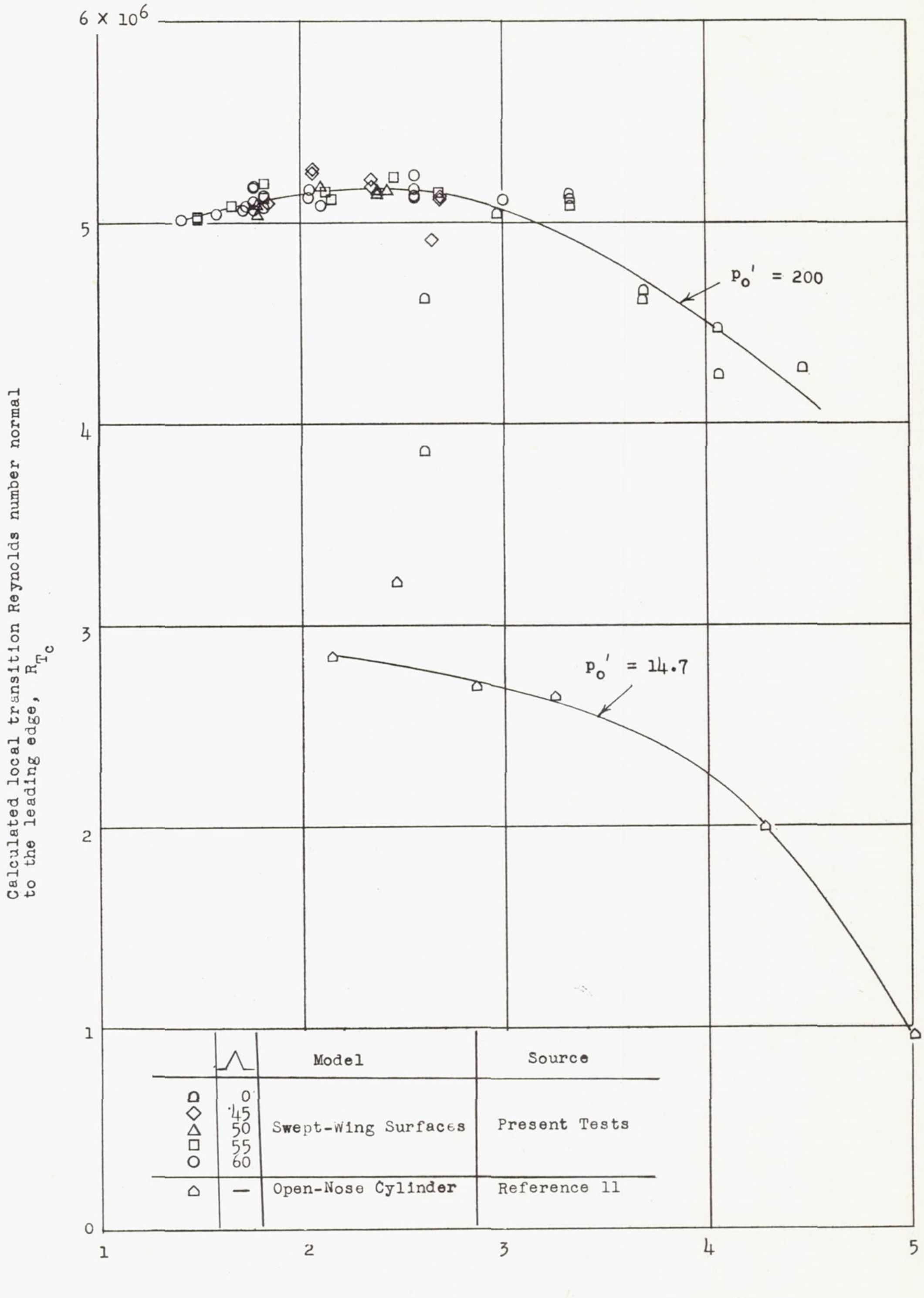


Figure 14.- Variation of transition Reynolds number with Mach number for a constant stagnation pressure.

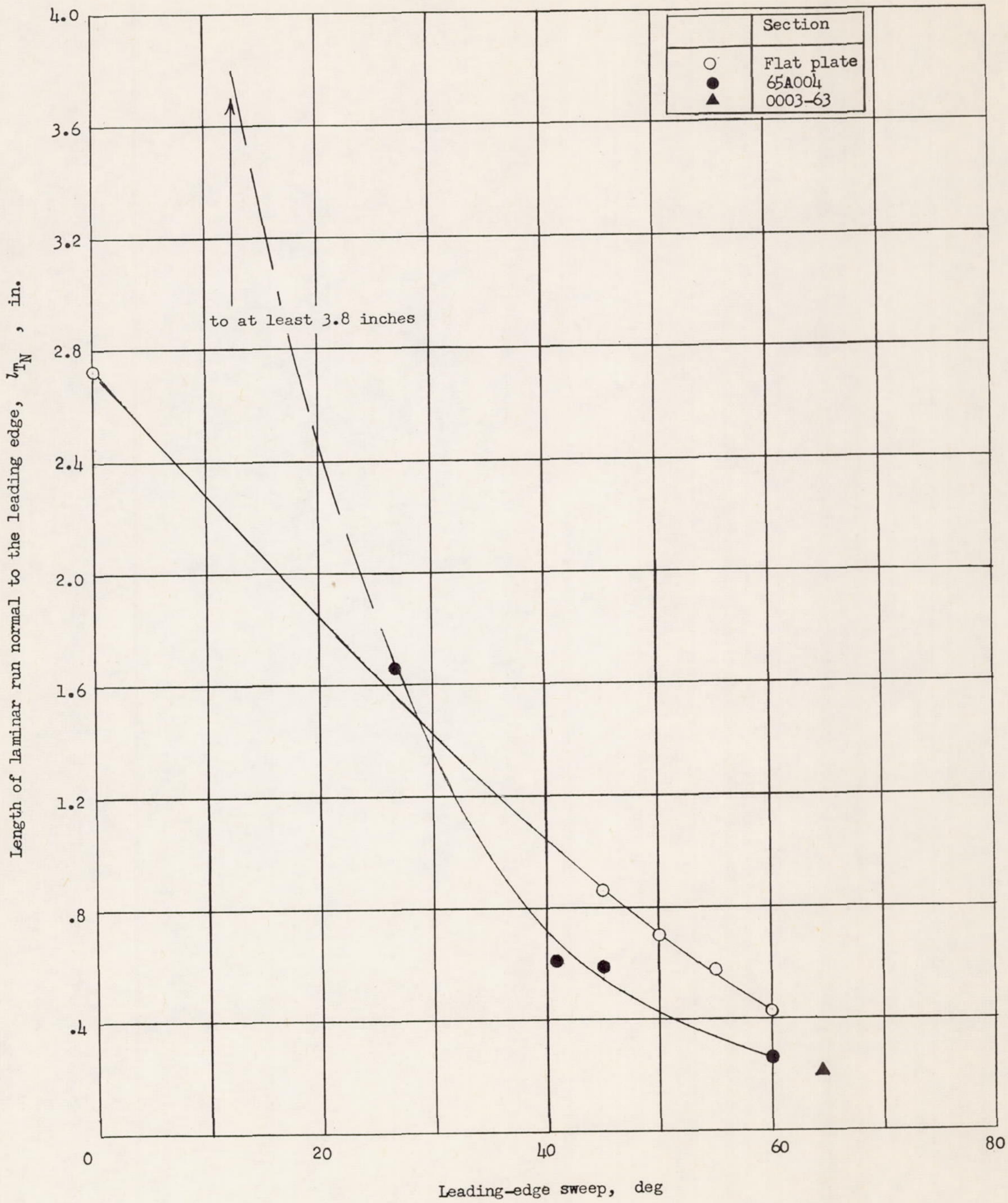


Figure 15.- Effects of wing section on boundary-layer transition as a function of leading-edge sweep angle at zero angle of attack. M = 4.04.